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RESEARCH MEMORANDUM

INVESTIGATION AT A MACH NUMBER OF 1.90 OF A DIVERTER-TYPE BOUNDARY-LAYER REMOVAL SYSTEM FOR A SCOOP INLET

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INVESTIGATION AT A MACH NUMBER OF 1.90 OF A DIVERTER-TYPE BOUNDARY-LAYER REMOVAL SYSTEM FOR A SCOOP INLET

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SUMMARY

An experimental investigation was conducted at a Mach number of 1.90 to determine the effect of a diverter-type boundary-layer removal system on the performance of a scoop inlet. The supersonic portion of the inlet consisted of a two-dimensional, reverse Prandtl-Meyer turn followed by a constant-area throat. A fuselage installation was simulated by mounting the inlet on a flat plate. The boundary-layer removal system consisted of a thin flat plate to split off the boundary layer and a wedge to divert the flow around the inlet. The distance between the splitter and boundary-layer plates was variable.

It was found that the inlet would not start completely. The pressure gradient at the corner apparently separated the small boundary layer which developed on the splitter plate itself, thereby causing a shock to be positioned at the leading edge of the plate. Appreciable spillage of air and loss in recovery resulted. Removal of the splitter plate permitted starting and resulted in satisfactory operation. Maximum pressure recovery and weight flow ratio were 0.86 and 0.96, respectively.

INTRODUCTION

The "scoop"-type side inlet is characterized by the fact that its supersonic compression surface is located outboard of the fuselage and deflects the flow toward the fuselage. This orientation potentially eliminates the high cowl drag normally associated with external compression inlets while maintaining the possibility of high pressure recovery.

The scoop-type inlet was first suggested formally by Rae in reference 1. In this investigation the full potential was not realized since it was impossible to fully start the inlet; this condition

resulted from shock-induced separation of the fuselage boundary layer combined with the starting problem inherent in the inlet type. The necessity for boundary-layer control was again demonstrated in reference 2 in which a total-pressure recovery of 0.80 at a Mach number of 1.9 was obtained with boundary-layer removal compared with 0.74 without. In reference 3, high recoveries are reported with an inlet designed for a Mach number of 2.7. Best recoveries were again obtained with boundary-layer control. It has thus become apparent that the scoop inlet generally requires boundary-layer control even though the boundary layer does not flow onto the compression surface. 2864

The present investigation was conducted before the publication of references 2 and 3, although the results of reference 3 were available. The purpose was to investigate a scoop inlet designed to operate with a simple diverter-type boundary-layer removal system. A similar removal system has since been reported in reference 2. However, since the system of reference 2 was not modified to satisfactory form, the limited data of the present investigation are being published as a guide to such correction.

The investigation was conducted at the NACA Lewis laboratory on an isentropic scoop inlet designed for operation at a Mach number of 1.9. A flat plate was employed to simulate a fuselage.

INLET DESIGN

A sketch of a rectangular scoop inlet is given in figure 1 for the purpose of defining the design variables. Air enters at a Mach number M_0, is turned through an angle θ_l by means of an oblique shock, and is then compressed isentropically to a Mach number M_1. (Symbols used herein are defined in the appendix.) The total turning angle is θ_c. A normal shock occurs at the Mach number M_1 and subsonic diffusion follows. The leading edge of the side plates can be swept back to lie in the plane of the oblique shock, since for operation with the normal shock swallowed no compression occurs ahead of this plane. The lower lip of the inlet is set a distance h above the fuselage for the purpose of boundary-layer removal.

For operation with the shock swallowed the inlet may have considerable contraction. Starting is accomplished by spilling air transversely between the fuselage and the side plate. In figure 2, for example, the shock is located in front of the inlet. The higher pressure in the region behind the shock causes air spillage through area ABC, thereby permitting the shock to move back toward the throat.

The various design variables and the factors which they affect are as follows: (a) Lip angle θ_l. For $\theta_l = 0$, the compression will

be isentropic. The compression surface will, however, be relatively long with a resultant thick boundary layer. As θ_l increases the length of surface decreases, but the pressure loss through the oblique shock increases. Therefore, an optimum wedge angle will presumably exist for each design Mach number. (b) Final Mach number M_l. For highest potential pressure recovery, $M_l = 1$. However, for $M_l = 1$ the air is turned away from the axial direction by the greatest amount thereby aggravating conditions further downstream where the air must be turned again. In addition, the closer the design value of M_l approaches unity, the more difficult the starting problem and the longer the compression surface. Again an optimum should exist for each free-stream Mach number. (c) Height-to-width ratio. This parameter has an important effect on starting. The amount of air which can be spilled depends on the height squared, whereas the amount which must be spilled for starting depends on the product of the height and width; consequently, the greater the height-to-width ratio, the greater the relative ability to spill air during the starting process.

For the inlet of the present investigation the free-stream Mach number was 1.90. A design having the greatest potential pressure recovery was employed. Accordingly, the wedge angle was chosen to be zero. The final Mach number was chosen as 1.30. The resultant turning angle and contraction ratio were 17.4° and 1.47, respectively (see fig. 3(a)). The leading edge of the side plate was swept back at the Mach angle, 31.8°.

For height-to-width ratio a value of 2.0 was selected. With this value a simplified calculation in which viscous effects were neglected showed that the inlet should start even if the flow coefficient for transverse spillage was as low as 0.3.

A constant-area throat section of 1.4 hydraulic diameters was included for shock stabilization.

The plate used to simulate a fuselage was 5 inches wide and extended 11 inches forward of the corner of the inlet. A 1/4-inch-wide strip of carborundum was placed 1/4 inch from the leading edge. Under conditions of the tests the thickness of the undisturbed portion of the boundary layer was 0.18 inch at the corner of the inlet.

Boundary-Layer Control

In the tests of reference 3, boundary-layer removal was accomplished by applying suction to a slot in the fuselage immediately ahead of the corner. In the present investigation the simpler diverter-type system was used. The system consisted of a short flat plate to split the flow and a wedge-shaped diverter. (See fig. 3(a).) The splitter plate extended 1.5 inches upstream of the corner.

Subsonic Diffuser

The length of the subsonic diffuser was chosen equal to that of an equivalent conical diffuser having a total included angle of 5°. The offset between the center line of the inlet and that of the subsonic diffuser outlet was limited by tunnel installation considerations to 3.5 inches and the chief design problem was selection of turning and diffusion rates to meet this limitation. Two diffusers were designed, the first (diffuser 1 in fig. 4) having rapid initial diffusion so that turning would occur at low speeds, the second having fairly uniform 2864
deceleration. The diffusion rate was varied by attaching inserts to the side walls. Both diffusers had the profile given in figure 3(b). Unless otherwise noted, data are for diffuser 1.

TEST FACILITY

Conditions. - The investigation was performed in the 18- by 18-inch tunnel of the Lewis laboratory. Tunnel Mach number from previous calibration was 1.90. Test-section total temperature and pressure were approximately 145° F and atmospheric, respectively, resulting in a Reynolds number of approximately 3.22×10^{6} per foot. The dewpoint was maintained at about -5° F.

Instrumentation. - Wall static-pressure distribution was obtained from taps located at various axial stations along both the supersonic and subsonic portions of the inlet. Total-pressure recovery and velocity profile after diffusion were obtained from a 13-tube rake located 5 inches downstream of the transition piece. Mass flow was measured by a calibrated sharp-edged orifice.

DISCUSSION OF RESULTS

Starting characteristics. - The starting characteristics of the inlet itself were determined by conducting the first tests without the simulated fuselage. The results are presented in figure 5. It can be seen that the inlet as designed would not completely start; the maximum mass flow ratio which could be obtained was 0.852. The peak recovery was 0.780 at a flow ratio of about 0.76. In the schlieren photograph of figure 6(a), it is evident that a strong shock existed at the leading edge of the splitter plate which accounted for the low values of recovery and flow ratio. This inability of the inlet to swallow the shock appeared to be an effect of one or both of two possible causes. First, the amount of contraction resulting from the choice of a throat Mach number of 1.3 may have been too great when combined with boundary-layer effects and possible separation of the flow at the corner; that is, the inlet may have been choking just

downstream of the corner. Second, the sudden compression at the corner of the inlet may have been great enough to separate the boundary layer on the splitter plate.

So that the effect of the amount of contraction could be checked, perforations were added at the throat and slots were cut in the side plates just back of the corner. Total bleed area was about 15 percent of the throat area. The effect on the flow is shown in figure 6(b) and the performance is given in figure 5. Because of the increased flow the shock at the leading edge of the plate moved slightly rearward. Peak pressure recovery increased about 2 percent. However, since the strength of the shock was still sufficient to cause appreciable spillage, starting had not been accomplished.

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Apparently the leading-edge shock and the resultant inability to start had been associated mainly with separation of the boundary layer on the splitter plate. Accordingly, the perforations and slots were filled in and the length of the plate was reduced in a stepwise manner. For each plate length the shock positioned itself at the leading edge with the result that each length reduction produced an increase in both pressure recovery and flow ratio. Best performance was obtained with the entire splitter plate removed. Maximum recovery and flow ratio were better than those of the original inlet by about 7 and 10 percent, respectively (see fig. 5). The schlieren photograph of figure 6(c) (pressure recovery, 0.748; flow ratio, 0.954) shows that the inlet is effectively started. The shock which stands just ahead of the corner results from the fact that the lower surface is inclined to the flow at an angle which is close to the maximum angle for an attached shock.

Effect of Fuselage Position

The effect of the position of the boundary-layer plate on the inlet without the splitter plate is given in figure 7. Both pressure recovery and flow ratio are relatively insensitive to plate position for spacings as low as 0.28. For the larger spacings, conditions are, of course, those for a nose inlet. For the smaller spacings, however, starting must be accomplished by transverse spillage. The schlieren photograph of figure 8 represents operation at a spacing of 0.28. It can be seen that the leading shock which was associated with the splitter plate has now been swallowed. The plate curves downward just ahead of the corner. This curvature accelerates and turns the flow in this region, thereby alleviating the detached shock condition which existed when the inlet was tested with the splitter plate removed. In addition, the curvature tends to cancel the shock emanating from the inlet lip.

It should be noted that in the absence of the splitter plate the amount of boundary layer actually entering the inlet cannot be determined directly from the spacing parameter because the boundary layer tends to follow the plate curvature. For a spacing of 0.28 the actual distance between the inlet and the plate at the corner is 1.5 boundary-layer thicknesses. In figure 8 the curvature of the boundary layer can be seen but the amount which actually enters the inlet cannot be ascertained.

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Subsonic Diffuser Performance

The theoretical recovery of the inlet neglecting skin friction effects is the total-pressure ratio across a normal shock at $M = 1.3$, that is, 0.979. With about 5 percent loss allowed for the subsonic diffuser, the pressure recovery should be about 0.93; the best experimental recovery was 0.86. The difference could have resulted either from a throat length which was insufficient for full normal shock diffusion (see, for example, ref. 4) or from too great an initial diffusion rate. Each of these could cause separation and local regions of high velocity.

The wall pressure distribution for the top surface of the inlet is presented in figure 9. A theoretical curve for zero subsonic diffuser losses is included for purposes of comparison. The theoretical pressure ratio across a normal shock at $M = 1.3$ is 1.80, whereas that observed experimentally for diffuser 1 was 1.62 or 90 percent of theoretical. The theoretical pressure rise in the subsonic diffuser was 1.46 and the experimental was 1.37 or 94 percent of theoretical. Evidently one fault was insufficient throat length.

Because of the manner in which the inlet was mounted in the tunnel, an increase in the constant-area throat length was impossible. It was possible, however, to decrease the amount of initial diffusion, thus effectively increasing the length of the throat. The theoretical Mach number and wall pressure variations of the redesigned diffuser, diffuser 2, appear in figures 4 and 9, respectively. In figure 9 it can be seen that the throat pressure ratio did improve; the ratio became 95 percent of theoretical. The pressure fell, however, in the first part of the subsonic diffuser and the subsonic pressure rise was only 84 percent of theoretical.

One reason for the poor performance of diffuser 2 can be found in figure 10. The static- and total-pressure distributions obtained from a rake located 11 inches downstream of the end of the diffuser are plotted for each diffuser. While the distributions for diffuser 1 are good, indicating maximum and minimum Mach numbers of 0.24 and 0.17, respectively, those for diffuser 2 are poor, indicating 0.40 and 0.08, respectively. The losses due to separation more than offset the gains in throat performance with the result that the average pressure recovery dropped to 0.82.

SUMMARY OF RESULTS

1. For a scoop-type inlet the pressure gradient which exists at the corner is great enough to separate even the thinnest of boundary layers with the result that the boundary layer must be removed immediately ahead of the corner if the inlet is to be completely started.

2864 2. A diverter-type boundary-layer removal system will operate satisfactorily with the scoop inlet.

3. The throat of the inlet must be of sufficient length to permit full shock diffusion.

4. If the offset between the center line of the inlet and that of the subsonic diffuser outlet is limited, indications are that better performance can be obtained by rapid initial diffusion followed by turning rather than by turning and diffusing simultaneously.

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APPENDIX - SYMBOLS

The following symbols are used in this report:

A	area
h	distance of corner of inlet above boundary-layer plate
M	Mach number
P	total pressure
p	static pressure
w	weight flow
w_0	free-stream weight flow through area equal to scoop area
δ	boundary-layer thickness
θ	turning angle

Subscripts:

0	free stream
1	after supersonic diffusion
2	after subsonic diffusion
c	at corner
l	at lip
r	rake
w	diffuser wall

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2. Dailey, C. L., Douglass, Wm. M., and McFarland, H. W.: Preliminary Investigation of Scoop Type Supersonic Diffusers. USCAL Rep. 11-1, Aero. Lab., Univ. Southern Calif., Dec. 15, 1951. (Summary Rep. USN BuAer Contract NOa(s) 12044.)

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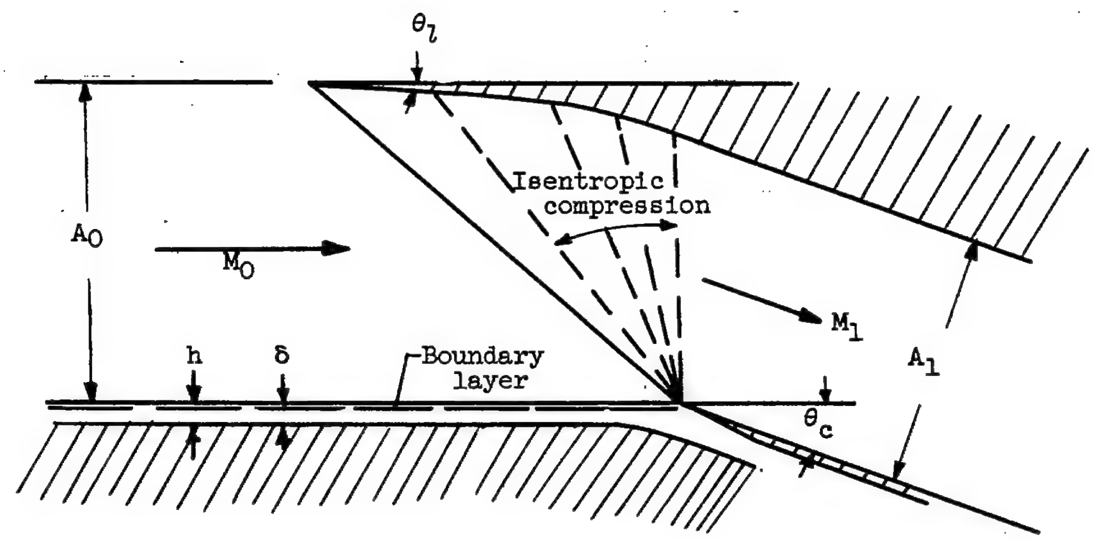


Figure 1. - Scoop inlet.

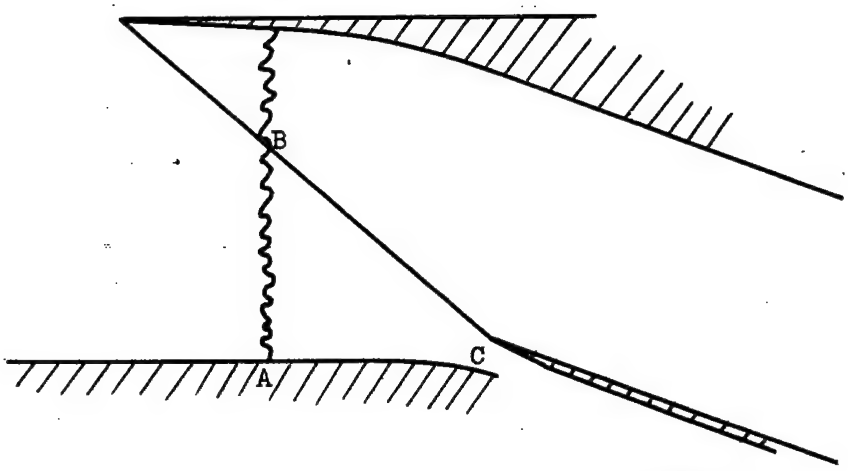
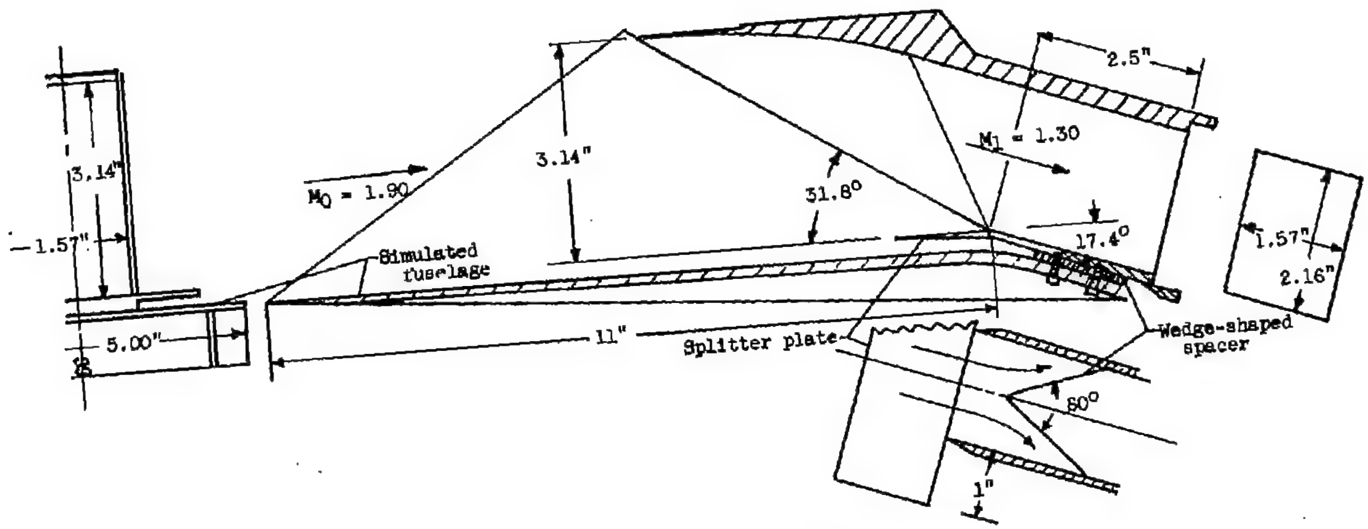
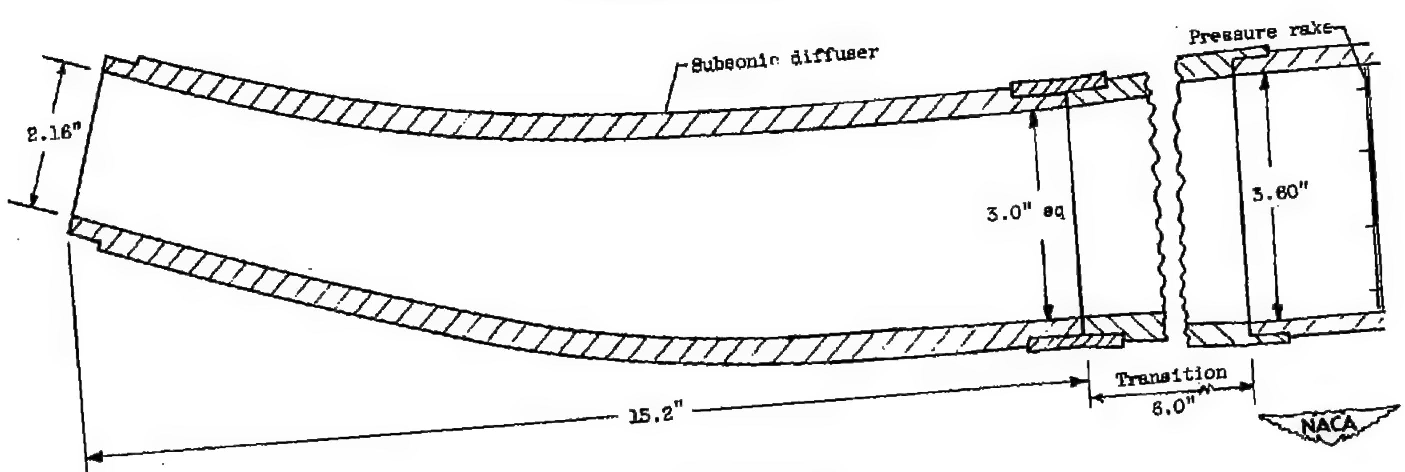


Figure 2. - Starting condition.

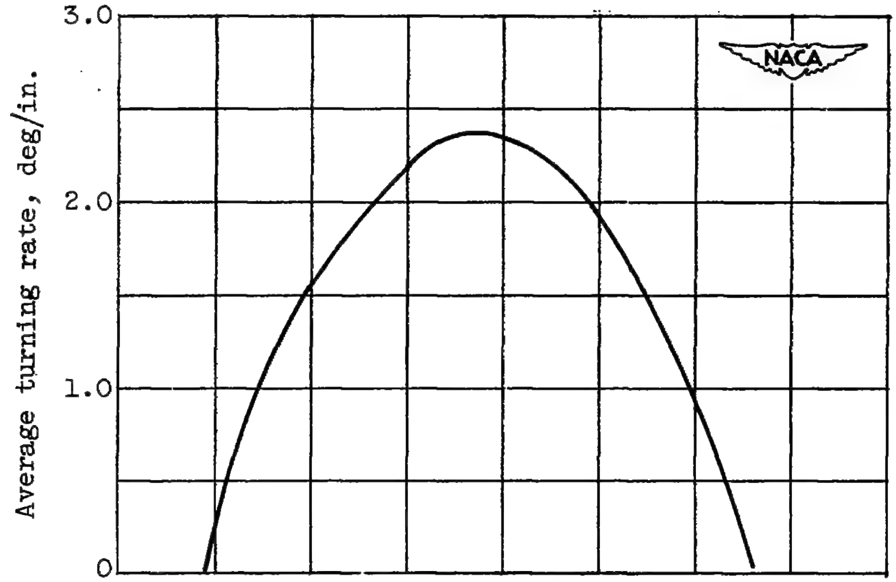


(a) Supersonic section.

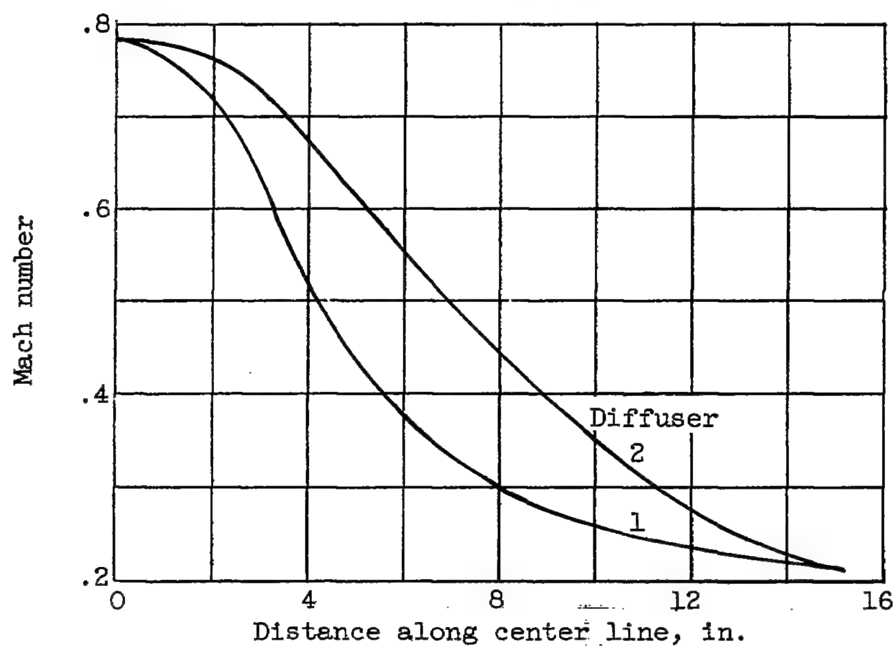


(b) Subsonic section.

Figure 3. - Inlet configuration.



(a) Turning rate.



(b) Mach number.

Figure 4. - Turning rate and Mach number distributions for subsonic diffusers.

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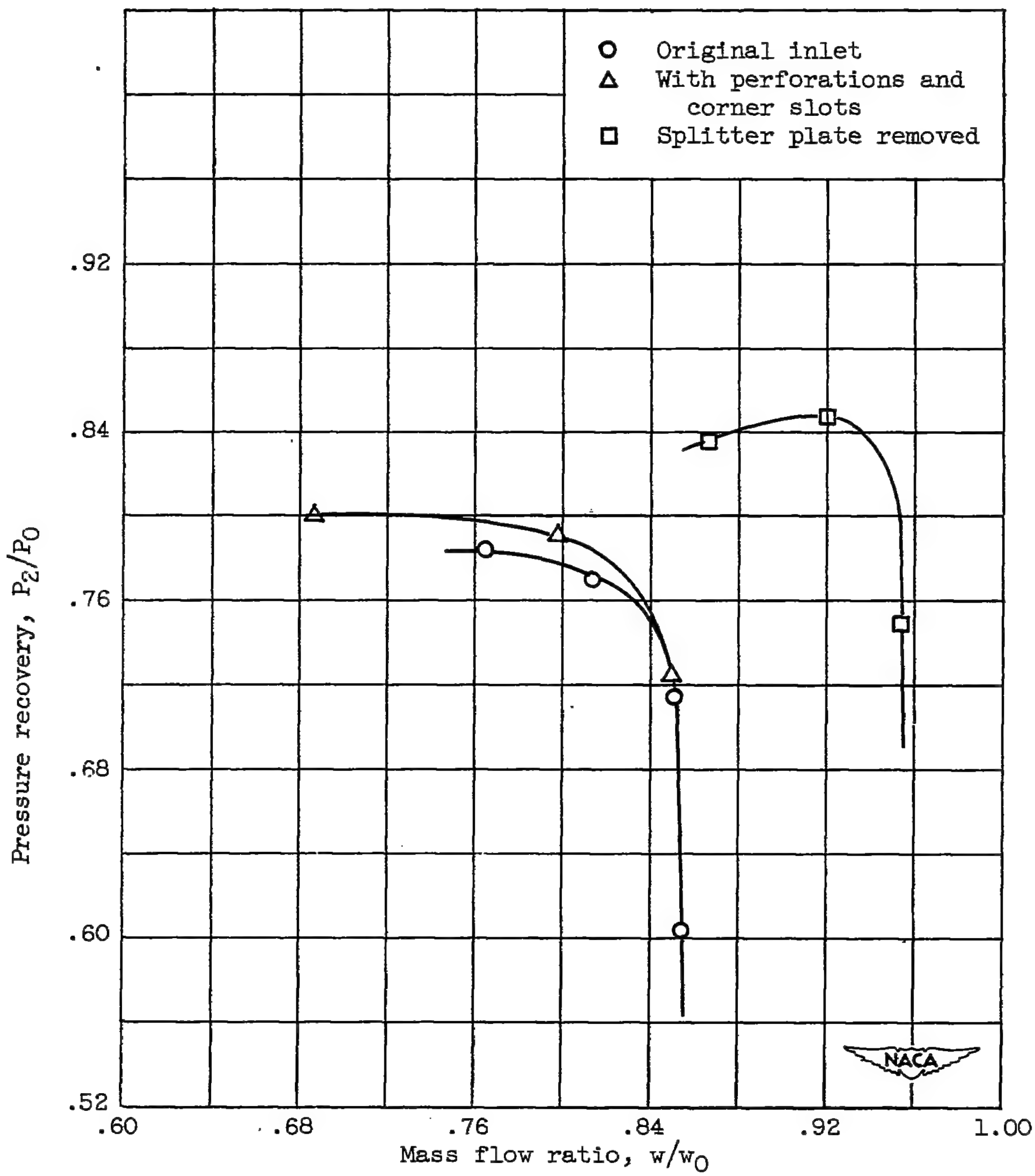

Figure 5. - Effect of splitter plate on inlet performance.

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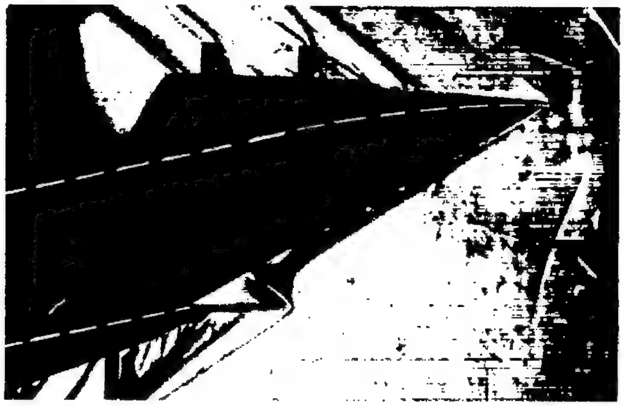
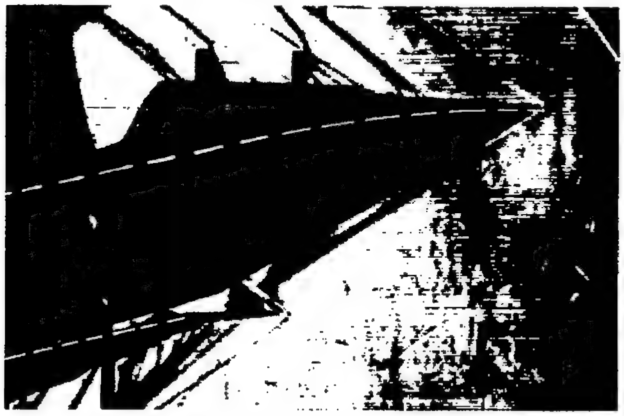
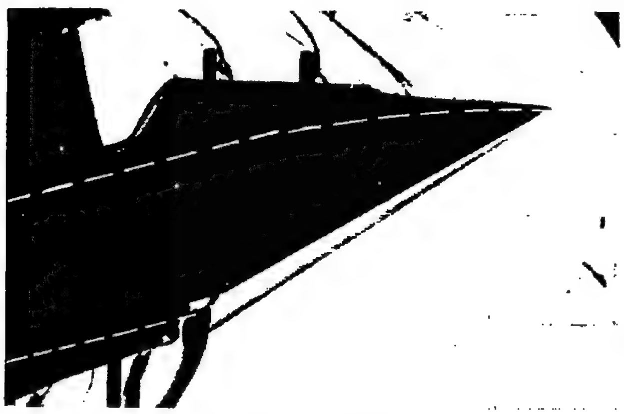


(a) Original inlet;pressure recovery, 0.715; flow ratio, 0.850.



(b) With perforations and corner slot; pressure recovery, 0.724; flow ratio, 0.848.



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(c) Splitter plate removed; pressure recovery, 0.748; flow ratio, 0.954.

Figure 6. - Effect of splitter plate on flow entering inlet.

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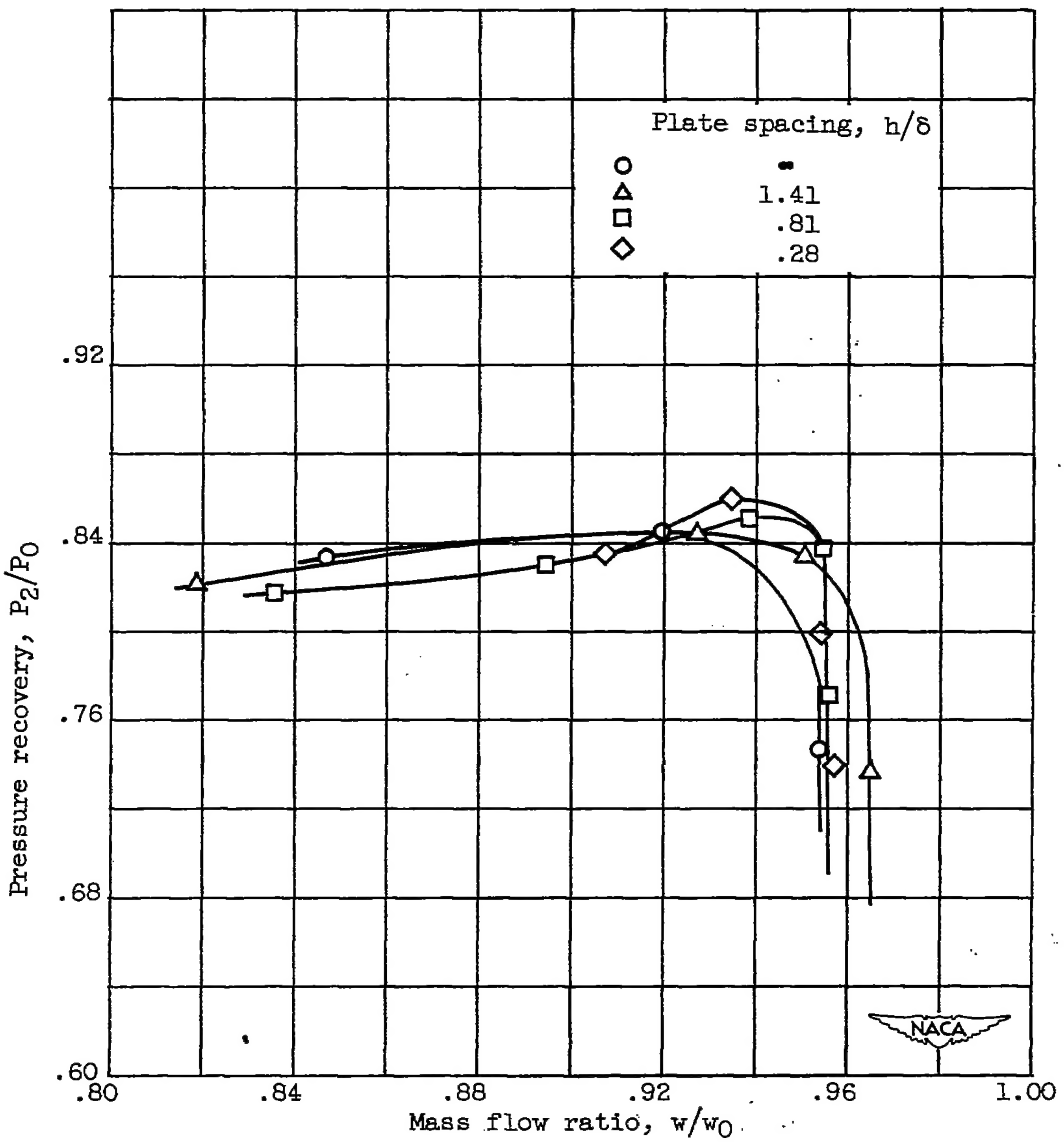

Figure 7. - Effect of position of boundary-layer plate on inlet performance.

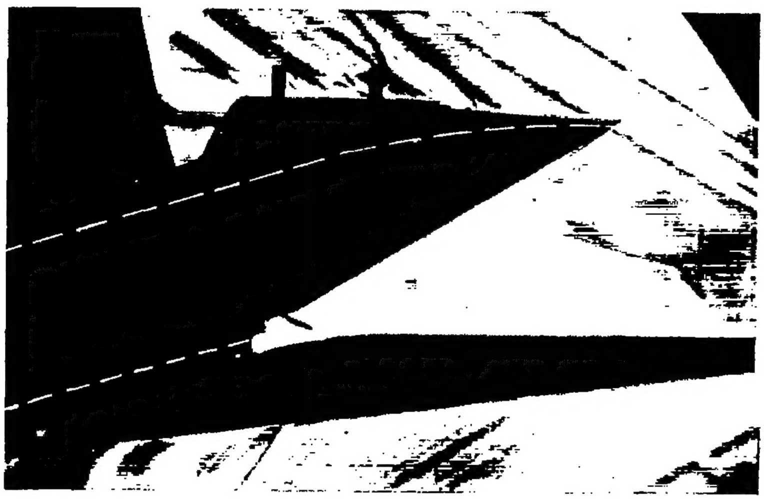

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Figure 8. - Effect of boundary-layer plate on flow entering inlet; plate spacing h/δ, 0.28; pressure recovery, 0.800; flow ratio, 0.955.

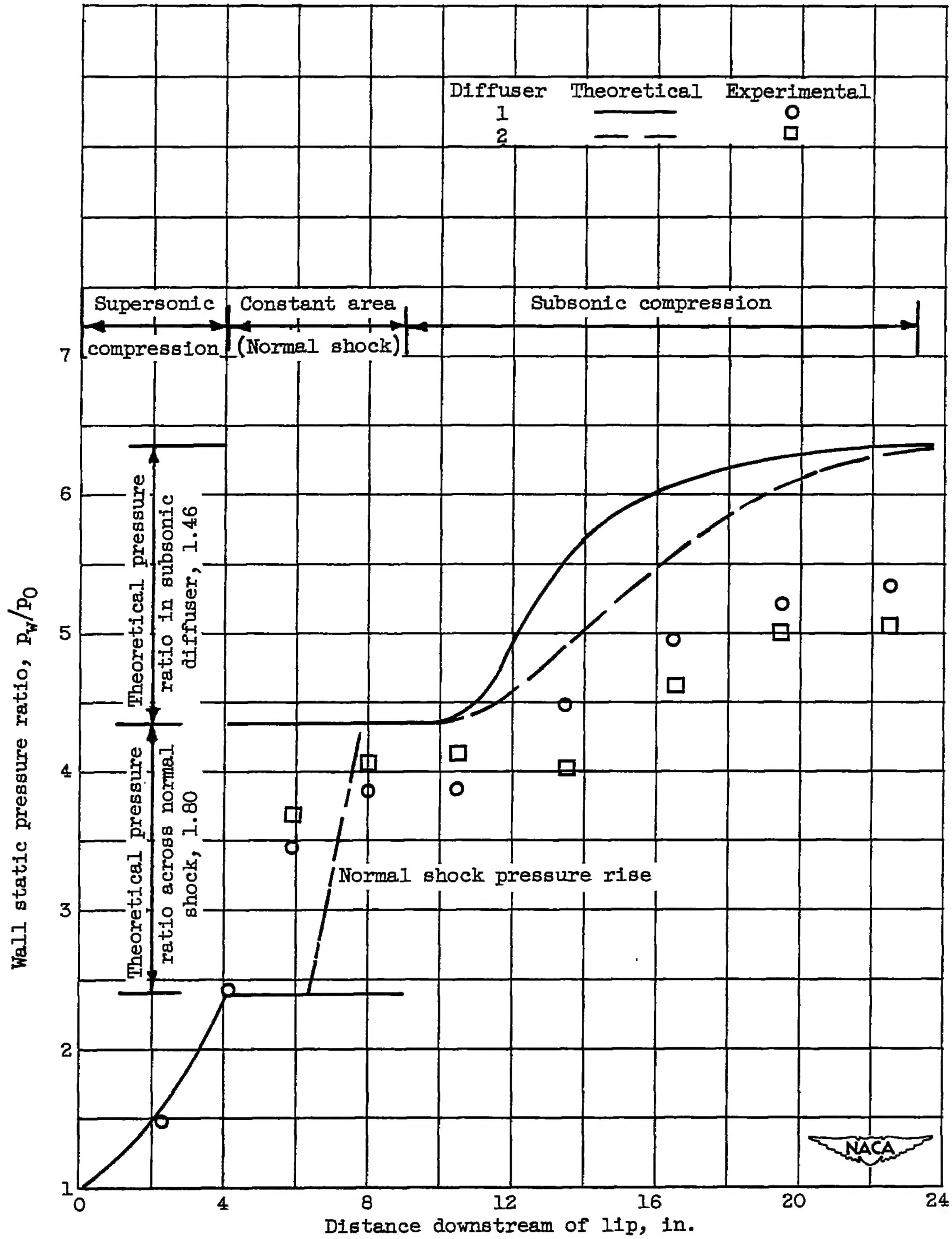

Figure 9. - Wall pressure distribution.

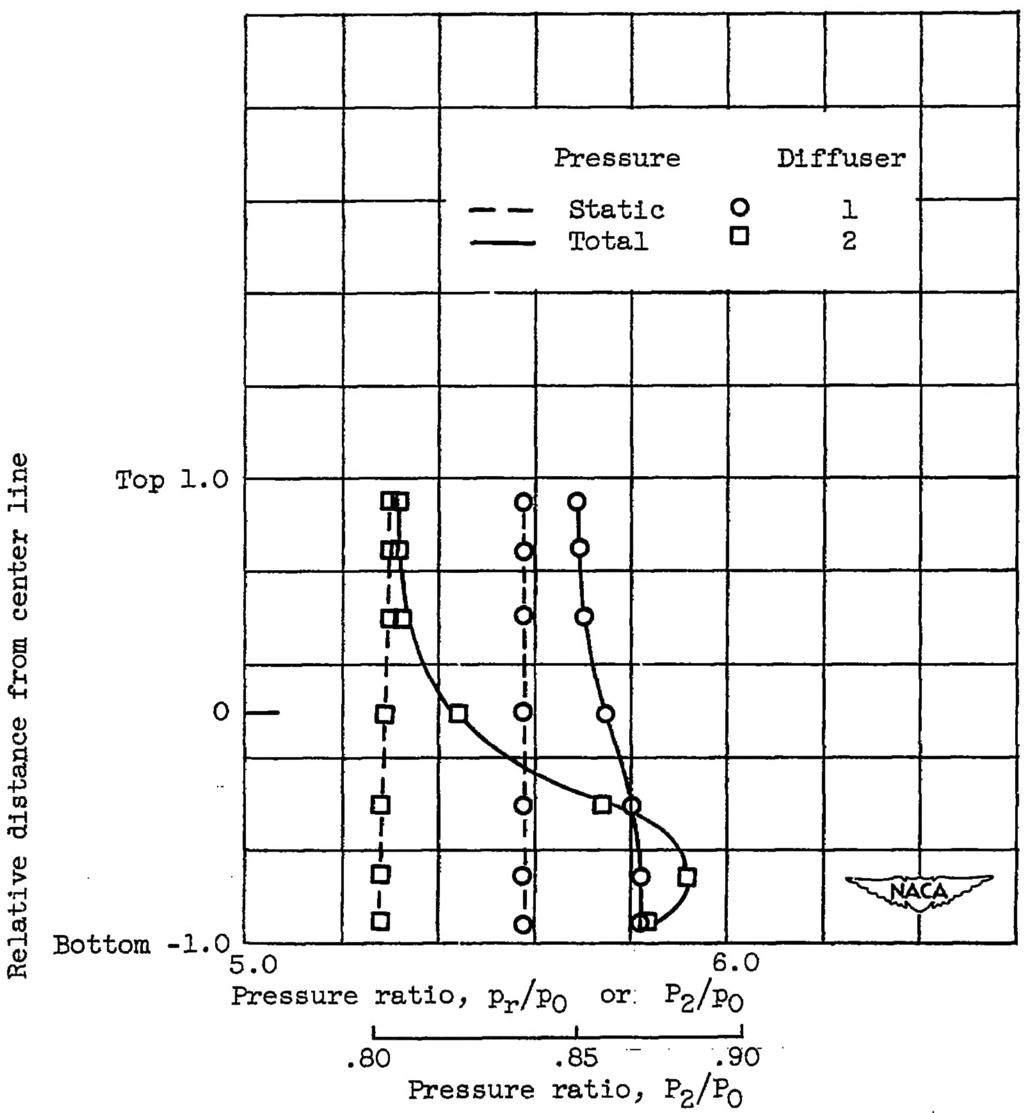

Figure 10. - Pressure distribution after diffusion.

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